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Lewis Research Center
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TECHNICAL PAPER proposed for presentation at
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Abstract

The SERT II spacecraft was launched February 3, 1970 after 3-1/2 years of development and testing. A 15-cm diameter mercury electron bombardment ion thruster onboard the spacecraft has operated for over 3 months of the 6-month mission goal at the time of this writing. The overall thruster efficiency has remained constant at approximately 0.68. The specific impulse is about 4200 sec., and the thrust is 28 mN (6.3 mlb) for a thruster input power of about 850 W. A second, redundant, flight thruster was operated briefly in space. In this paper the performance of both flight thrusters is compared with corresponding preflight ground testing. In addition, the endurance of one flight thruster is compared with two ground endurance tests of prototype thrusters.

Introduction

Space Electric Rocket Test I (SERT I) in 1964 was the first NASA test of an ion thruster in space.(1) The SERT I flight verified the production of thrust and the neutralization of an ion beam in space. SERT II (Fig. 1) was launched February 3, 1970 into a near polar orbit of 1000 km for the purpose of demonstrating long life (6 months) space operation of either one of two ion thruster systems on board. The spacecraft is powered by a nominal 1.5 kw solar cell array. Companion papers describe the design of the spacecraft, the SERT II mission, the auxiliary experiments performed, and the development of the power conditioner.(2-5) The extensive developmental thruster ground testing, which established the confidence necessary for the flight, has been reported.(6-8)

This paper presents and compares data taken with the flight thrusters, both in prelaunch ground tests and in orbital flight. The results presented include flight data through June 12, 1970. (The oral conference presentation will include results through August 1970.) The data include electrical parameters which allows evaluation of the thruster power efficiency and durability of various thruster components. The flight thruster propellant utilization efficiency has been estimated, based upon ground test results, as no means were included in the SERT II system for direct measurement of propellant flow. The thrust, determined by three different methods, is presented and discussed in a companion paper.(9)

Thruster and Spacecraft Description

The SERT II thruster is a nominal 1 kW power level, 15-cm diameter mercury electron-bombardment ion thruster. The mercury ion thruster was invented by Harold R. Kaufman of the NASA Lewis Research Center. Fig. 2 is a cutaway view of a SERT II thruster which shows the various components such as propellant reservoirs, discharge chamber, and ion accelerator grid. Both the thruster and neutralizer propellant reservoirs contain an estimated 8 month (5800 hr) supply of mercury. For the purpose of this paper "thruster" is defined to include all components shown in Fig. 2. The thruster and power conditioner together are defined as a "thruster system".

In operation, liquid mercury is fed by positive displacement (nitrogen gas behind a butyl rubber diaphragm) to a porous tungsten vaporizer plug. The vaporizer plug is heated to pass a controlled amoung of mercury vapor through the electrical isolator. (The component shown in Fig. 2 is a simulated isolator made of stainless steel. It is the approximate size and thermal equivalent of a ceramic isolator.) An experiment to test an electrical isolator in the main propellant line was not developed in time to meet the flight schedule.

Approximately one-third of the mercury vapor from the isolator flows through the thruster cathode. The remainder flows through a flow splitting orifice, into the distributor, then into the main discharge chamber. A 1.6 to 2.0 amp discharge is drawn from the hollow cathode to the main anode and, through interaction with the mercury propellant vapor, creates a mercury plasma within the 15-cm diameter discharge chamber. Permanent bar magnets located around the chamber improve the ionization efficiency of the discharge. The baffle and magnetic-field shaping pole pieces produce desirable characteristics of the discharge.(8) Ions in the plasma discharge diffuse to the screen-accelerator grids, where they are extracted and focused into a 0.25 amp , 3000 v mercury ion beam. The ion beam is neutralized by an equal current of electrons injected from a hollow cathode neutralizer. The mercury for the neutralizer cathode discharge is supplied by a separate feed system, similar to, but smaller than, the main feed system. Details of the construction of the neutralizer vaporizer and cathode are provided in reference 7. A ground screen encloses the entire thruster except for the accelerator grid and prevents ambient plasma electrons from streaming to thruster components at positive high voltage. A more complete description of the thruster and its general operation is given in references 6, 7, and 8.

Two thruster systems are installed on the SERT II flight spacecraft as shown in Fig. 3. Thruster 1 is oriented so as to raise the spacecraft orbit and thruster 2 is oriented so as to lower the spacecraft orbit. One thruster system is a backup in the event of failure of the other system. The solar array produces only enough power to run one thruster system at a time. The solar array output voltage and current are switched directly to the input of the operating power conditioner. The high voltage outputs of the power conditioner are unregulated and vary directly with the solar array output voltage. Solar array voltage may vary with time due to long-term degradation or seasonal sun-array angle changes. Any such array voltage variation will cause corresponding changes in the thruster specific impulse, thrust, and overall

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thruster efficiency.

The SERT II spacecraft is gravity gradient and control-moment-gyro stabilized and a 2 degree or greater thrust misalinement will cause spacecraft tumbling. Initially the thrusters are properly alined, but launch vibration, thermal distortion (most notably that of the grid system), or accelerator grid wear could change the thrust vector direction. Therefore, each thruster has a gimbal system which permits realinement of the thrust vector. Each gimbal system consists of an inner ring which was an integral part of the thruster structure, an outer ring, two actuators and pin pullers. (To date, use of the gimbal system has not been required.)

Each flight thruster, fully loaded and including the gimbal system has a mass of 30 kg which breaks down as follows,

	kg
Gimbal system	7.7
Thruster	3.0
Propellant	
Main tank	14.1
Neutralizer tank	0.9
Tankage and Supports	4.3
	30.0

Thruster Test Procedure

Flight Operation

A Thorad/Agena launch vehicle injected the SERT II spacecraft into a near circular near polar orbit of 1000 km altitude. After a planned series of spacecraft maneuvers and operational confirmations, stand-by thruster (thruster 2) operation was initiated on February 10, 1970. It was operated for a planned duration of two days to confirm correct operation in space and was then shut down. The primary thruster (thruster 1) operation was initiated on February 14, 1970, and it has been operating to this time (June 12, 1970) with two brief interruptions. (It was shut down for about 17.5 hours during the solar eclipse of March 7, 1970. Also, on May 21, 1970 the thruster system shut down for about 10 hours because of excessive high voltage cycling.)

In start-up, the thrusters were preheated for a minimum of 1-1/2 hours to insure that mercury condensation did not occur in any portion of the thruster or feed system. Preheat power was supplied by resistive heating of the main cathode, the propellant isolator, and the neutralizer cathode. The neutralizer keeper discharge was also established during preheat to stabilize its operation and provide a source of electrons when the ion beam was initiated. After preheat, the thruster discharge was initiated without the extraction of an ion beam (high voltages not on). The thruster discharge current was controlled by a feedback loop with the thruster vaporizer. The thruster system then came to near-equilibrium operating temperatures and thrust could be produced when the high voltages were turned on. Upon extraction of an ion beam, the main vaporizer control automatically switched to the normal control mode that held a constant ion beam current. Stable thrust levels were achieved with either thruster in less than one minute.

As mentioned previously, there was a chance that a badly misalined thruster would cause the SERT II spacecraft to tumble. This imposed a mission constraint on the thrust initiation sequence. Thrust was required to be turned on and stabilized while the spacecraft was in direct communication with a ground station. In addition, the thrust level was required to be low enough to allow timely thrust vector correction with the gimbal system if necessary. The total time available while in communication with a given ground station was about 15 minutes, which dictated that the thrust level be stabilized within a few minutes.

The ion beam was first turned on at a nominal 30 percent of full thrust (actually about 11 mN). This low level of thrust insured that any probable disturbance torques would be small enough to be controllable with the gimbal system. The thrust was then increased to 80 percent of full thrust (about 22 mN) in a single step, and then to 100 percent of full thrust (about 28 mN). As mentioned previously, gimbal adjustment was not required at any thrust level.

The thruster system (thruster and power conditioner) is self-protecting and does not require continuous monitoring from ground tracking stations. Isolated electrical overloading of the high voltage power supplies causes automatic recycling to clear the overload source. If continuous recycling exists for approximately 100 sec the thruster system is automatically shut down and must be restarted by ground command. If the power conditioning input voltage from the solar cell array is not within a specified range the thruster system also shuts down and requires a ground restart command.

Flight Thruster Ground Testing

Both flight thrusters were operated in a series of three ground tests before flight. Test 1 calibrated the thruster propellant flows (no propellant tanks used) and discharge characteristics over a greater range of thruster operation than expected in flight. Both neutralizer and main flow were measured by timing the fall of mercury in capillary tubes. The power supplies contained flight-like high frequency inverter circuits, but with more range flexibility than the flight power conditioner. The test was made in a 1.5 m diameter tank with a stainless steel target 4.5 m from the thruster. The test time was limited to less than 25 hr to minimize condensed back sputtered target material.

Test 2 verified correct operation of the finally assembled thruster with the flight power conditioner. Accurate thruster electrical performance data were obtained by reading the thruster system telemetry outputs on a digital voltmeter. This test was made in the same 1.5 m diameter tank as test 1 and was limited to less than 15 hours thruster operation.

Test 3 was made after the flight thruster had been mounted on the flight spacecraft, and had been subjected to vibration and thermal-vacuum tests as part of the total spacecraft system. This test verified correct thruster system operation on the flight spacecraft and was the last test prior to flight. For this test the SERT II command station at Lewis Research Center was used to initiate

commands and to receive telemetered data. The test was conducted in a 4.5 m diameter vacuum tank with a stainless steel target located 10 m from the thrusters. Each thruster was operated for approximately 50 hr. The spacecraft, including the non-operative thruster, was covered with a movable shield to minimize condensed back sputtered material.

Ground Endurance Tests

Prior to the flight, two ground endurance tests were initiated at contractor facilities using thruster systems identical to those on the flight spacecraft. The vacuum facility for each test contains a frozen mercury target located 1.6 m from the thruster. The use of a mercury target greatly reduces the amount of condensed backsputtered material on the thruster. In one test, Test M conducted at McDonnell Company (St. Louis, Mo.) the power conditioner is contained in the same tank as the thruster. In the other test, Test T conducted at TRW Systems, Inc. (Redondo Beach, Calif.), the power conditioner is contained in a small, separate vacuum tank.

Data Accuracy

During the flight and the flight qualification test (Test 3) telemetered data was received in a quantized form called counts. A range of 0 to 61 counts corresponded to the full range for each measured parameter. The possible error due to this quantizing was combined with all other possible errors (including power conditioner calibrations) by the method of root sum of squares of the errors. The resulting calculated error is listed in the last column of Table I.

For the first thruster calibration test (Test 1), 1 percent meters were used for beam current, discharge voltage, and discharge current. Three percent meters were used for the balance of measurements. The propellant mass utilization measurements (using capillary flow tubes) are reproducible to about 1 percent for the main flow and about 5 percent for the neutralizer flow.

Thruster Performance

Tables I and II list the ground and flight performance of thrusters 1 and 2, respectively, under conditions of preheat, propellant but no beam and 30, 80, and 100 percent thrust. The data of Tables I and II will be discussed in detail in this section. Table III summarizes thruster efficiencies for the flight thrusters and the ground endurance test thrusters.

Flight Thruster I

Preheat. - The neutralizer cathode to-keeper discharge lit properly in all attempts in flight. The time to establish the discharge depends on the thermal time constant of the cathode and vaporizer, and the time required for cathode activation. The time to light the neutralizer was 6 to 12 min in ground tests and was about 4 min in flight. The shorter lighting time in flight was attributable to a higher neutralizer system initial temperature about 55° C in space compared to 25° C for ground tests. A higher thruster system temperature was caused by a combination of factors which include: a skew sun-spacecraft angle that permitted more solar flux to impinge in the thruster area, slight changes of some thermal control surfaces, and the Agena spacecraft interface which was not present during ground tests. Once lit, the neutralizer cathode discharge was stabilized by its closed-loop control which held a constant neutralizer keeper voltage via a feedback loop with the neutralizer vaporizer heater. Thruster system 1 remained in preheat for 1.5 hr in ground tests and 1.8 hr in the flight. The longer flight preheat was a result of the times between ground station coverage for the flight. The neutralizer vaporizer heater current (item 12, Table I) was about 0.3 amp lower in flight than in ground tests. This occurred both in preheat and subsequent operation and is probably the result of the hotter thermal environment in space. The test 1 neutralizer vaporizer heater current value, 2.60 amp, was typical of a new start (which requires more neutralizer flow). The test 1 power conditioner input voltage was maintained at 60 v, the expected mean value for 100 percent thrust operation in space.

The next preheat step, initiation of the main discharge with no beam extraction, was accomplished without problems in flight or ground tests. (This mode of operation was not possible in test 1 with the power supplies used.) In flight, the main discharge initiated within 0.8 min. Typical ground test discharge initiation times were from 0.5 to 2 min. The thruster cathode power cut back as programmed once the main discharge current exceeded 0.3 amp. After lighting, the main discharge stabilized within one minute at its normal level of two amps. The main discharge current was controlled via a feedback loop with the main vaporizer. Thruster system 1 remained in this propellant mode for 1.6 hr to allow thermal stabilization of the thruster.

Operation at reduced thrust. - As stated previously, the thruster was operated at the nominal 30 percent thrust level in order to assess spacecraft stability. As shown in Table I, the flight thruster operation at 30 percent thrust was similar to that of the ground tests. Propellant flow rates were not measured during test 1 at 30-percent thrust because this mode was not planned as a space endurance operating point; the sole purpose was to avoid introducing a spacecraft tumble mode in the event of a major thrust vector misalinement. Other ground tests indicated 50 to 60 percent propellant mass utilization efficiency at 30-percent thrust.[6]

The thruster was at 30 percent thrust for 3.2 hr during flight. This time was alotted to analyze spacecraft stability and send gimbal adjustments if necessary. The thrust vector was found to be alined within one degree of the center of mass of the spacecraft, however, and no gimbal adjustment was required

The thrust was next increased to the 80 percent level and the neutralizer keeper voltage set point was changed from 28 to 23 v. The 80 percent thrust set point was provided to enable the thruster to operate in the event of severe thruster and/or solar cell degradation. The lower neutralizer keeper voltage set point was used for endurance running because ground tests showed longer neutralizer lifetimes at the lower voltage.[7] The higher neutralizer keeper voltage set point was used for initial operation because of

improved neutralizer control stability.

The 80 percent thrust point stabilized within 3 minutes in flight and was operated for 1.9 hr. The ion beam set point (0.20 amp) and the neutralizer set point (22.9 v), were as expected. Flight operation was in good agreement with ground tests (Table 1).

Operation at full thrust. - The beam current set point was changed to full thrust (0.25 amp) and the main discharge voltage set point was lowered to 37 v from the 40 v set point. The value of 37 v was chosen to enhance main cathode lifetime on the basis of many developmental lifetests.(7)

The arrival at full beam in space was accomplished without incident. The thruster stabilized within 2 min at an operating point almost identical to ground tests. The ion beam current of thruster system 1 was as expected within the accuracy of the telemetry system. An occasional flight reading has been recorded at the next higher quantized telemetry value of 0.258 amp.

The level of the main discharge current was of concern for it strongly influences the main cathode lifetime.(7) The discharge current is sensitive to many parameters, such as, ratio of cathode to main flow, screen to accelerator grid spacing, level of beam current, level of beam extraction voltage, strength of magnetic field, and ambient (tank or space) gas density. In flight, the initial discharge current value of 1.67 amp was in excellent agreement with the ground tests. This value of discharge current rose slowly with time. This variation is further discussed in the section on Thruster Endurance.

Most other electrical parameters of thruster 1 were similar in space to those measured in ground tests (Table 1, 100 percent thrust). The thruster and neutralizer vaporizer power required in space were slightly lower, possibly because of the hotter thermal environment on the spacecraft. The screen current (I_5) equaled the neutralizer emission current (I_9) within the accuracy of the measurements. The net ion beam current was equal to the screen current minus the beam current impinging on the accelerator grid. The impinging beam current consisted of both direct and charge exchange ions, but was not the sole contributor to the accelerator current (I_6). Ground tests showed that 0.5 ma of the accelerator current was due to ions created by the neutralizer discharge falling back to the accelerator grid. Therefore, the net ion beam current was computed as ($I_5 \times 10^3 - I_6 + 0.5$) ma.

The level of accelerator current for thruster 1 in space was equal (within data accuracy) to that measured in tank tests. This result indicates that there is negligible charge exchange accelerator impingment produced by interactions of the beam ions with the facility background gas, at the background pressure of 5×10^{-6} torr.

The potential difference between the neutralizer cathode and local ground (tank wall or space plasma potential) is presented in table 1. For ground tests with thrust being produced the value was -9v to -18v while for space tests it was -19v to -36v. This voltage is a function of the neutralizer discharge flow rate, the shape and magnitude of the ion beam, and, during ground tests, the tank geometry. On the SERT II flight this voltage was also dependent on orbital position.(4) This so-called floating voltage represents a loss in ion energy and hence must be considered in calculations of thrust and beam power.

The calculated performance of both flight and ground test thruster systems is summarized in table III. The mass utilization efficiency includes the neutralizer flow. For flight thruster 1, the overall thruster efficiency in space and ground tests was approximately constant at 0.68. However, the flight power efficiency will decrease with operating time as the positive high voltage decreases (decrease is directly proportional to solar cell degradation). The estimated flight mass utilization efficiencies were based on measured flight electrical parameters and ground propellant flow rates. The propellant flow rates used for comparison were those obtained on the ground for the flight thruster and other thrusters operated over a similar range of electrical parameters.

As previously noted, on March 7, 1970, the SERT II spacecraft twice passed through the moon's penumbra during a solar eclipse. There was insufficient solar flux at these times to supply thruster operating power. The thruster was shut down before the first pass through the eclipse and turned on 17.5 hrs later after the second pass through the eclipse. Shutdown and startup were normal and without incident and all thruster parameters returned to their previous values. Again, no thruster gimbaling was required.

High voltage breakdowns. - The high voltage breakdowns of the thruster system were monitored throughout the flight. Breakdowns can occur in either the power conditioner or thruster. However, it seems more likely that breakdowns will occur in the thruster because of events such as accelerator grids sputtering and building up condensed material which might precipitate a breakdown. A breakdown is defined as a current overload of either the positive or negative high voltage power supply. If a current overload occurs, both high voltage supplies are shut down for 0.1 sec and then automatically turned on again.(5) In some cases, the overload clears itself in one shutdown cycle. Other times the overload may cycle several hundred times (10 to 20 sec) before clearing.

The overall number of high voltage electrical breakdowns of the thruster system in space was about the same as experienced in ground testing. During initial thrusting operation in space, however, there were fewer than expected. During ground tests a large number of breakdowns are usually experienced in the first hour of thruster operation. The time between breakdowns then lengthens over the first 50 hr of operation to a period of about 1 to 10 hr per breakdown. In space, however, there were no electrical breakdowns for the first eight hours of thruster 1 high-voltage operation. Following this 8 hr there were 38 breakdowns in the next 50 hr. The period between breakdowns then lengthened to an average (for the 2800 hr reported herein) value of about 8 hr. The mean period between breakdowns (for the same 2800 hr) was 6 hr. The maximum period observed between breakdowns was 60 hr, but typically, periods were scattered with 70 percent of the breakdowns having a period between 3 and 14 hr. At 2385 hr the thruster system sustained high

voltage overload cycling for about 100 sec. This amount of recycling caused an automatic shutdown of the thruster. Ground commands were necessary to restart the system. The restart preheat, 30, 80, and 100 percent thrust steps, were accomplished without incident. The cause of the sustained overload cycling is not known. The period between breakdowns following the restart was somewhat shorter than typical, there being 20 arcs in the 50 hr period following the restart.

Thrust measurements. - A comparison of measured thrust is presented in detail in a companion paper.[9] Three different measurement methods agree to within the data system error. The 100 percent thrust level (approximately 28 mN or, 6.3 mlb), calculated using electrical parameters, is presented in Table III of this paper. This calculated thrust is uncorrected for beam divergence or multiple ionization. The thrust was also determined by an accelerometer and an orbit raising technique. The initial orbit of 1000 km has been raised by thruster 1, to 1068 km at the time of this writing.

Flight Thruster 2

Thruster system 2 is the backup thruster system for the SERT II mission and was operated prior to thruster system 1 in order to verify operation and check the thrust vector alinement. The total thruster system 2 space operating time was 41.6 hr. Flight thruster system 2 was operated using the same procedure as described for thruster system 1. The times to initiate and stabilize the neutralizer and main discharges and beam current were nearly identical (less than 0.5 min difference) for the two flight thrusters. There were 14 high voltage electrical breakdowns during the 41.6 hr operating time. As with thruster system 1, no gimbal adjustments were necessary.

The performance of thruster 2, both in ground and flight tests, is listed in Table II for various modes of operation (preheat, etc.). The procedure and accuracy of obtaining data for Table II are the same as that for Table I (thruster 1). The ground and flight data are in good agreement with the exception of the main discharge current. The indicated main discharge current in flight was 0.2 amp higher than in ground tests. The reason for this difference is unknown. Other minor variations between ground and flight results are attributable to the same causes as given for flight thruster system 1.

The calculated performance of thruster system 2 is summarized in Table III. The calculated thruster efficiency after 10 hr of space operation was 0.66 for thruster 2 as compared to 0.68 for thruster 1. The difference was attributed to a lower power efficiency because of a higher indicated main discharge current. Also, the mass utilization efficiency was lower because of a higher estimated neutralizer flow consumption at the higher level of neutralizer keeper current.

Ground Test Thrusters

Two ground endurance tests of prototype thruster systems were started before the space flight and are continuing as of this writing. The purpose of the tests is to accumulate data on component durability that may be correlated with flight test performance. Earlier tests were also made at McDonnell Company (St. Louis) and TRW Systems, Inc. (Redondo Beach), the two contractors at which the present tests are being conducted. The tests were started in December 1969 after incorporating a design change to the main vaporizer. The main cathode was also replaced at that time because it and the main vaporizer are welded together as one assembly. Other components, including the neutralizer system, the accelerator grids, and the power conditioner, however, were reused. The hours on the various thruster system components prior to the start of the present tests are listed below.

	Test M, hr	Test T, hr
Thruster cathode-vaporizer	45	44
Neutralizer system	1207	919
Accelerator grid	1207	1534
Thruster body	1207	1534
Power conditioner	1140	1245

Tables IV and V give the thruster system performance results which are discussed in detail in the following sections. The hours presented for test M and test T, on Tables IV and V respectively, are only those hours accumulated since the final design change (November 1969) on the prototype thruster. The thrusters of test M and test T are in every way identical in design to the flight thrusters.

Test M. - Prior to the start of endurance testing all components of the test M thruster were successfully vibrated at 50 percent higher levels than expected for launch. The electrical performance was unchanged by the vibration tests and was, in general, the same at the start of endurance testing as it was during test 1. Minor variations were due to differences in input voltage and power conditioners used. The calculated thruster efficiencies, shown in Table III, are similar for test 1 and the life test. At the end of the life test the residual propellant in the neutralizer and main propellant storage tanks can be weighed and an average propellant flow rate will be determined.

Minor interruptions occurred to the life test at 410 and 1700 hr when temporary loss of facility pumping occurred. There was, however, no loss of vacuum and the life test resumed after a normal preheat cycle.

It is seen from Table IV that for the life test, all thruster parameters with the exception of the neutralizer heater voltage and current remained nearly constant up to hour 2740. Over this time period the neutralizer vaporizer-cathode heater current showed a decline from 1.98 to 1.68 amp. This type of long-term decline has not been observed in any other endurance test including the flight. At 2740 hr into test M (3947 hr total on neutralizer system) the neutralizer vaporizer control loop began to oscillate. Although the thruster-neutralizer still functioned, that is, produced a normal steady beam, the thrust level was changed to the 80 percent value where the neutral-

izer control loop became stable. The corresponding thruster efficiency and operation at 80 percent thrust are shown in Tables III and IV, respectively. The thruster has operated at 80 percent thrust from 2750 hr to the time of this writing. Neither the cause of the long-term decline in neutralizer vaporizer current nor the control loop oscillation are presently known. Detail observation of the neutralizer cathode is not possible without opening the vacuum tank.

Test T. - The performance of the thruster during test 1 and test T is shown in Table V. The hours accumulated on the thruster system components prior to the start of test T were listed previously. The operation of the thruster during test 1 and the start of test T were similar. As with test M, minor differences were obtained due to differences in power conditioners, facility pressure, and input voltage. The power conditioner input voltage was progressively lowered from 66 to 60 v to simulate estimated solar cell degradation. This decrease in input voltage was in part responsible for the variation of the thruster parameters with time (Table V). In particular, this decrease in input voltage caused the discharge current to increase at a faster rate than in test M.

The life test was interrupted at 480, 1400, and 2440 hr by temporary facility problems. There was no appreciable loss in vacuum at the shutdowns occurring at 480 and 2440 hr, however, and the life test was resumed after normal thruster preheat. The facility problem at 1400 hr necessitated exposure of the thruster to the atmosphere. The life test was restarted after cleaning the thruster grids and discharge chamber of condensed sputtered material.

At hour 2886 the neutralizer propellant reservoir unexpectedly became empty and the life test was stopped. Prior to this test, the thruster system had been tested for about 919 hr at LeRC on a prototype spacecraft. Subsequent analysis of that test showed that neutralizer flow rates were twice as great as expected. This higher flow was probably due to many restarts and long preheat periods while on the prototype spacecraft. For test T, the average equivalent mercury flow rate for the 2886 hr was 29 ma compared to a predicted 21 ma, or about 38 percent higher. No reason is presently known for the higher consumption while on life test. The life test is planned for restart after reloading the neutralizer reservoir and cleaning the thruster of condensed sputtered material.

Thruster Endurance

At the time of this writing, June 12, 1970, (2800 hr into the flight test) there is no evidence of any serious degradation of thruster components. In general, flight data indicates that the degradation is equal to or less than that indicated in ground life testing. Of course several thruster components are unaffected by operation in the sense that they experience no degradation. These components include the thruster body, anode, and propellant distributor. The strength of the permanent magnet field and the flow calibration of vaporizers have been found to be invariant with thruster operating time. All electrical insulators are shadow shielded against condensed deposits and show no measurable electrical leakage. Heater wires and their insulation show no measurable evidence of deterioration in flight or in either ground life test.

The parts of a thruster that have been found to wear and have lifetime limits are the accelerator grid and the hollow cathodes. The accelerator grid wear was discussed in references 6 and 7 wherein it was shown that no problem for the 6-month duration of the mission was expected. Thruster cathode erosion is indicated by an increase of the thruster discharge current with time. Variation of discharge current with time for the flight thruster and the thrusters of tests M and T is shown in Fig. 4. The general trend is an increase in discharge current with time. The flight data, being quantized, appear as straight lines. The regions of overlap are for times when the quantized reading alternated between two values and the actual value was probably in between.

The discontinuities in the ground test discharge current trends are caused by test interruptions. The exact influence of these interruptions on cathode lifetime is not known. The near vertical portion of the curves for 50 hr following either a restart or the initial start may be related to the pump-down of the facility and minor thermal warping of the accelerator grids. Facility gas molecules can diffuse into the discharge chamber, causing a change in the discharge characteristics. A high facility pressure tends to cause a lower discharge current. The equilibrium position of the accelerator grids is different between start (cold) and operation (hot). The cold grid spacing is somewhat closer, and a closer grid spacing lowers the discharge current required to produce a given beam current.

As is seen from Fig. 4, the overall rate of increase in discharge current for the flight is less than that of either ground test. The increase of discharge current with time in test T is greater than that of test M. This difference, as previously explained, is felt to be due, in part, to the decrease of the power conditioner input voltage with time in test T. For test M and the flight test, the input voltage remained essentially constant.

The limit on cathode lifetime is actually a limit built into the power conditioner which saturates at a discharge current of 2.5 amps. However, the end of thruster life is not necessarily reached when the discharge current reaches 2.5 amp. Two options are available: (1) the discharge voltage can be commanded to a lower level (35 v) which will reduce the discharge current or (2) the beam current can be commanded to the 80 percent thrust level where the discharge current is much lower (1.3 amp). The neutralizer cathode is not subject to the same type of lifetime problem. Because its flow is controlled, the neutralizer discharge can be adjusted for minor cathode wear variations. (A neutralizer cathode, run under simulated conditions, has operated for over 10,800 hr.[7])

Concluding Remarks

Both thrusters on the SERT II flight have, to date, operated as expected in space. They passed the stresses of launch vibration and space thermal-vacuum. The backup thruster was first tested for 41.6 hr to verify its operation. Then the primary thruster was turned on and has operated at 100

percent thrust for over 3 months of its 6-month goal at the time of this writing. It was not found necessary to gimbal either thruster. Thruster system electrical breakdowns have averaged approximately one every 8 hr on the flight and have been with one exception automatically cleared by the control used. There is no indication of wear on any component except the main cathode and that wear rate is less than the lowest wear rate experienced in ground tests. The maximum operating time of the primary thruster appears at this time to be limited by an exhaustion of the propellant supply at approximately eight months rather than by thruster wear.

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TABLE I. - PERFORMANCE OF FLIGHT THRUSTER 1

Item Index (for tables I, IV, and V)			
Item	Description	Item	Description
1	Thruster vaporizer heater voltage, V_2, v	12	Neutralizer vaporizer and cathode heater current, I_7, amp
2	Thruster vaporizer heater current, I_2, amp	13	Neutralizer keeper voltage, V_8, v
3	Thruster cathode heater voltage, V_3, v	14	Neutralizer keeper current, I_8, amp
4	Thruster cathode heater current, I_3, amp	15	Neutralizer cathode to tank wall (or to space) voltage, v
5	Discharge voltage, V_4, v	16	Neutralizer emission current, I_9, amp
6	Discharge current, I_4, amp	17	Thruster cathode keeper voltage, V_{10}, v
7	Positive high voltage (screen), V_5, v	18	Thruster cathode keeper current, I_{10}, amp
8	Screen current, I_5, amp	19	Pressure of thruster environment, torr
9	Negative high voltage (accelerator), V_6, v	20	Thruster equivalent neutral flow rate, amp
10	Accelerator current, I_6, ma	21	Neutralizer equivalent neutral flow rate, amp
11	Neutralizer vaporizer and cathode heater voltage, V_7, v	22	Power conditioner input voltage, V_{in}, v

Item	Preheat				Propellant, no beam			30% thrust (11 mN)			
	Test 1[a]	Test 2[b]	Test 3[c]	Flight[d]	Test 2	Test 3	Flight	Test 1	Test 2	Test 3	Flight
1	0	0	0	0	2.67	1.51	1.32	(f)	2.55	1.32	1.17
2	0	0	0	0	1.67	1.63	1.52	1.73	1.73	1.52	1.33
3	>15	15.9	15.7	15.7	8.00	7.20	0.60	[f]4.0	7.50	6.80	7.20
4	2.80	2.88	2.92	2.88	1.43	1.47	1.47	1.50	1.43	1.47	1.47
5	50	>50	>50	>50	39.5	39.6	3.92	38.0	42.3	39.0	38.7
6	0	0	0	0	1.92	1.99	2.13	0.38	0.56	0.59	0.70
7	0	0	0	0	0	0	0	2980	3360	3470	3470
8	0	0	0	0	0	0	0	0.076	0.085	0.088	0.094
9	0	0	0	0	0	0	0	1580	1700	1790	1740
10	0	0	0	0	0	0	0	1.20	0.98	1.03	1.03
11	9.40	8.57	7.53	6.42	8.36	7.12	6.24	7.60	7.42	6.42	5.76
12	2.60	2.30	2.31	2.05	2.20	2.11	1.96	2.20	1.98	1.96	1.78
13	28.2	28.8	29.2	28.6	28.8	29.2	28.6	28.0	28.4	28.6	28.0
14	0.200	0.204	0.241	0.221	0.182	0.213	0.190	0.200	0.184	0.213	0.198
15	0	0	0	-6	0	0	-6	-9	(f)	(f)	-19
16	0	0	0	0	0	0	0	[f]0.075	0.085	(f)	0.094
17	300	>416	>416	>416	8.5	11.3	10.0	14.8	20.1	20.0	18.2
18	0	(h)	(h)	(h)	(h)	(h)	(h)	0.300	(h)	(h)	(h)
19	4×10^{-6}	3×10^{-6}	4×10^{-7}	10^{-10}	3×10^{-6}	4×10^{-7}	10^{-10}	6×10^{-6}	6×10^{-6}	4×10^{-7}	10^{-10}
20	0	0	0	0	(f)	(f)	(f)	(f)	(f)	(f)	(f)
21	(f)	(f)	(f)	(f)	(f)	(f)	(f)	(f)	(f)	(f)	(f)
22	[e]60	70	74	70	70	74	68	[e]60	66	67	68

Item	80% thrust (22 mN)				100% thrust (28 mN)							Telemetry uncertainty (rss)
	Test 1	Test 2	Test 3	Flight	Test 1	Test 2	Test 3	Flight				
								10 hr	1000 hr	2000 hr	2800 hr	
1	(f)	2.55	1.72	1.51	(f)	2.60	1.72	1.51	1.51	1.51	1.51	±0.07
2	1.93	1.93	1.71	1.63	1.96	1.88	1.78	1.71	1.71	1.71	1.71	±0.08
3	[f]4.1	7.50	6.80	6.80	[f]4.4	7.68	6.80	7.20	7.20	7.20	7.20	±0.35
4	1.50	1.43	1.47	1.47	1.50	1.43	1.47	1.47	1.47	1.47	1.47	±0.05
5	38.1	41.5	38.2	38.1	3.70	37.3	37.2	37.1	36.9	36.9	36.9	±0.2
6	1.10	1.05	1.04	1.10	1.68	1.70	1.57	1.67	1.76	1.76	1.80	±0.05
7	2980	2940	3190	3120	2980	2960	3070	3070	2930	2930	2930	±65
8	0.200	0.199	0.203	0.203	0.250	0.253	0.253	0.253	0.253	0.253	0.253	±0.005
9	1580	1540	1640	1590	1560	1530	1540	1540	1490	1490	1490	±50
10	1.30	1.15	1.14	1.25	1.28	1.57	1.49	1.60	1.49	1.60	1.49	±0.12
11	7.35	7.34	6.42	6.00	7.25	7.02	6.68	5.76	6.00	5.76	5.76	±0.25
12	2.15	1.98	2.05	1.91	2.11	1.87	2.01	1.80	1.86	1.80	1.80	±0.05
13	21.9	23.1	23.6	22.9	22.0	23.0	23.6	22.9	22.9	22.9	22.9	±0.7
14	0.200	0.177	0.198	0.193	0.200	0.176	0.193	0.190	0.182	0.185	0.185	±0.004
15	-14.7	(f)	(f)	-19	-17.5	(f)	-14	-27	-24	-27	-36	±2
16	[f]0.196	0.198	(f)	0.205	[f]0.242	0.249	(f)	0.258	0.258	0.258	0.258	±0.006
17	12.0	11.2	12.7	12.3	11.6	10.7	12.3	12.3	11.3	11.3	11.3	±0.5
18	0.300	(h)	(h)	(h)	0.300	(h)	(h)	(h)	(h)	(h)	(h)	(h)
19	5×10^{-6}	5×10^{-6}	4×10^{-7}	10^{-10}	4×10^{-6}	5×10^{-6}	4×10^{-7}	10^{-10}	10^{-10}	10^{-10}	10^{-10}	(h)
20	0.276	[g]0.241	[g]0.265	[g]0.265	0.314	[g]0.309	[g]0.315	[g]0.313	[g]0.313	[g]0.313	[g]0.313	(h)
21	0.027	[g]0.023	[g]0.025	[g]0.026	0.021	[g]0.016	[g]0.017	[g]0.017	[g]0.016	[g]0.017	[g]0.017	(h)
22	[e]60	60	63	63	[e]60	61	61	61	59	59	59	±1

[a]Calibration test in 1.5 meter diameter vacuum facility.
[b]Test in 1.5 meter diameter vacuum facility, fully assembled, loaded thruster and flight power conditioner; data from digital voltmeter values of hard-lined telemetry.
[c]Final preflight systems test on flight spacecraft in 4.5 meter diameter vacuum facility; data from telemetry.
[d]Flight data.
[e]Laboratory supplies made equivalent to this power conditioning input voltage.
[f]The accuracy of these values is questionable or data unavailable.
[g]Flow rate estimated from ground tests.
[h]No flight telemetry channel.

TABLE II. - PERFORMANCE OF FLIGHT THRUSTER 2

Item	Description	Item	Description
1	Thruster vaporizer heater voltage, V_2, v	12	Neutralizer vaporizer and cathode heater current, I_7, amp
2	Thruster vaporizer heater current, I_2, amp	13	Neutralizer keeper voltage, V_8, v
3	Thruster cathode heater voltage, V_3, v	14	Neutralizer keeper current, I_8, amp
4	Thruster cathode heater current, I_3, amp	15	Neutralizer cathode to tank wall (or to space) voltage, v
5	Discharge voltage, V_4, v	16	Neutralizer emission current, I_9, amp
6	Discharge current, I_4, amp	17	Thruster cathode keeper voltage, V_{10}, v
7	Positive high voltage (screen), V_5, v	18	Thruster cathode keeper current, I_{10}, amp
8	Screen current, I_5, amp	19	Pressure of thruster environment, torr
9	Negative high voltage (accelerator), V_6, v	20	Thruster equivalent neutral flow rate, amp
10	Accelerator current, I_6, ma	21	Neutralizer equivalent neutral flow rate, amp
11	Neutralizer vaporizer and cathode heater voltage, V_7, v	22	Power conditioner input voltage, V_{in}, v

Item	Preheat				Propellant, no beam			30% thrust (11 mN)			
	Test 1[a]	Test 2[b]	Test 3[c]	Flight[d]	Test 2	Test 3	Flight	Test 1	Test 2	Test 3	Flight
1	0	0	0	0	2.57	1.92	1.63	(f)	2.63	1.70	1.63
2	0	0	0	0	1.76	1.61	1.41	1.76	1.85	1.51	1.51
3	>15	15.9	16.0	16.0	9.38	9.07	8.66	[f]4.4	8.86	10.3	7.85
4	2.78	2.83	2.77	2.86	1.55	1.51	1.54	1.50	1.52	1.46	1.54
5	50.0	>50	>50	>50	40.8	40.2	39.9	41.6	42.5	42.3	42.2
6	0	0	0	0	1.67	1.88	2.00	0.40	0.55	0.66	0.71
7	0	0	0	0	0	0	0	2980	3500	3430	3490
8	0	0	0	0	0	0	0	0.074	0.080	0.088	0.088
9	0	0	0	0	0	0	0	1570	1730	1730	1730
10	0	0	0	0	0	0	0	0.91	1.15	1.13	1.13
11	8.70	8.90	8.88	7.70	8.90	8.66	7.70	8.70	8.41	7.48	6.60
12	2.45	2.40	2.43	2.33	2.40	2.38	2.28	2.43	2.20	2.08	1.98
13	28.2	27.8	28.5	28.5	27.8	28.5	28.5	21.3	27.9	28.5	27.8
14	0.200	0.195	0.206	0.226	0.195	0.186	0.199	0.200	0.182	0.210	0.215
15	0	0	0	-6	0	0	-9	-9.5	(f)	(f)	-24
16	0	0	0	0	0	0	0	[f]0.072	0.081	0.087	0.087
17	300	395	>416	>416	10.4	13.9	12.3	22.0	21.1	25.0	20.4
18	0	(h)	(h)	(h)	(h)	(h)	(h)	0.300	(h)	(h)	(h)
19	7×10^{-6}	3×10^{-6}	4×10^{-7}	10^{-10}	5×10^{-6}	10^{-10}	10^{-10}	4×10^{-6}	5×10^{-6}	4×10^{-7}	10^{-10}
20	0	0	0	0	(f)	(f)	(f)	(f)	(f)	(f)	(f)
21	(f)	(f)	(f)	(f)	(f)	(f)	(f)	(f)	(f)	(f)	(f)
22	[e]60	66	74	70	66.5	74	68	[e]60	70	67	68

Item	80% thrust (22 mN)				100% thrust (28 mN)				Telemetry uncertainty, (rss)
	Test 1	Test 2	Test 3	Flight	Test 1	Test 2	Test 3	Flight	
1	(f)	2.56	1.78	1.70	(f)	2.53	1.78	1.70	±0.07
2	1.85	1.90	1.70	1.70	1.93	1.97	1.77	1.77	±0.08
3	[f]4.8	8.97	10.7	8.25	[f]5.90	8.82	10.7	8.25	±0.35
4	1.50	1.49	1.46	1.54	1.50	1.48	1.46	1.54	±0.05
5	41.7	41.6	41.5	41.5	37.3	37.0	36.8	36.6	±0.2
6	1.19	1.14	1.21	1.21	1.77	1.72	1.83	1.98	±0.05
7	2980	3230	3160	3160	2970	2920	3030	3030	±65
8	0.200	0.195	0.203	0.203	0.250	[f]0.250	0.253	0.253	±0.005
9	1570	1640	1640	1640	1570	1490	1530	1530	±50
10	1.02	1.15	1.25	1.36	1.38	1.41	1.68	1.82	±0.12
11	7.85	8.36	7.70	6.38	7.50	8.05	7.48	6.38	±0.25
12	2.21	2.17	2.02	1.93	2.14	2.16	1.98	1.93	±0.05
13	21.0	23.6	24.0	24.0	23.1	23.6	24.0	23.4	±0.7
14	0.200	0.186	0.202	0.206	0.200	0.186	0.194	0.199	±0.004
15	-14.0	(f)	(f)	-35	-17.5	(f)	-14	-38	±2
16	[f]0.197	0.195	0.201	0.201	[f]0.242	0.242	0.254	0.254	±0.006
17	13.9	12.4	17.8	13.9	13.2	12.1	17.4	13.9	±0.5
18	0.300	(h)	(h)	(h)	(h)	(h)	(h)	(h)	(h)
19	5×10^{-6}	4×10^{-6}	4×10^{-7}	10^{-10}	5×10^{-6}	5×10^{-6}	4×10^{-7}	10^{-10}	(h)
20	0.252	[g]0.252	[g]0.252	[g]0.252	0.310	[g]0.311	[g]0.311	[g]0.312	(h)
21	0.027	[g]0.025	[g]0.025	[g]0.025	0.021	[g]0.021	[g]0.021	[g]0.021	(h)
22	[e]60	66	63	63	[e]60	61	61	61	(h)

[a]Calibration test in 1.5 meter diameter vacuum facility.
[b]Test in 1.5 meter diameter vacuum facility, fully assembled, loaded thruster and flight power conditioner; data from digital voltmeter values of hard-lined telemetry.
[c]Final preflight systems test on flight spacecraft in 4.5 meter diameter vacuum facility; data from telemetry.
[d]Flight data.
[e]Laboratory supplies made equivalent to this power conditioning input voltage.
[f]The accuracy of these values is questionable or data unavailable.
[g]Flow rate estimated from ground tests.
[h]No flight telemetry channel.

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TABLE III. - SERT II THRUSTER EFFICIENCY AND PERFORMANCE SUMMARY

Item	Description
1	eV/ion, discharge
2	Power eff.
3	Mass utilization eff.
4	Thruster eff.
5	Specific impulse, sec
6	Thrust, mN

Flight thruster 1, 100% thrust

Item	Test 1[a]	Test 2[b]	Test 3[c]	Flight data, telemetry			
				10 hr	1000 hr	2000 hr	2800 hr
1	212	215	194	208	222	222	227
2	0.89	0.88	0.90	0.89	0.89	0.89	0.88
3	0.75	0.77	0.76	0.76	0.77	0.76	0.76
4	0.67	0.68	0.68	0.68	0.68	0.67	0.67
5	4100	4210	4230	4240	4190	4130	4120
6	27.8	27.9	28.5	28.5	28.0	27.8	27.7

Flight thruster 2, 100% thrust

Item	Test 1	Test 2	Test 3	Flight data, telemetry
				10 hr
1	227	[d]217	230	250
2	0.87	0.88	0.87	0.87
3	0.76	0.76	0.76	0.76
4	0.66	0.66	0.66	0.66
5	4140	4110	4180	4180
6	27.8	[d]27.6	28.4	28.4

Test M thruster

Item	100% thrust					80% thrust	
	Test 1	Lifetest data, digital voltmeter					
		10 hr	1000 hr	2000 hr	2740 hr	2750 hr	4500 hr
1	236	243	258	261	269	208	208
2	0.88	0.89	0.89	0.89	0.88	0.90	0.90
3	0.75	0.77	0.77	0.77	0.77	0.63	0.63
4	0.66	0.68	0.68	0.68	0.68	0.57	0.57
5	4080	4260	4360	4330	4330	3600	3600
6	27.5	29.3	29.3	29.3	29.3	23.2	23.2

Test T thruster, 100% thrust

Item	Test 1	Lifetest data, digital voltemeter			
		10 hr	1000 hr	2000 hr	2886 hr
1	203	206	219	228	244
2	0.88	0.88	0.88	0.87	0.87
3	0.74	0.73	0.74	0.74	0.74
4	0.65	0.65	0.65	0.65	0.64
5	4080	4120	4150	4090	4020
6	27.8	28.6	28.8	28.4	27.6

[a]Calibration test in 1.5-meter diameter vacuum facility.
[b]Test in 1.5-meter diameter vacuum facility; data from digital voltmeter values of hard-lined telemetry.
[c]Final preflight systems test on flight spacecraft in 4.5-meter diameter vacuum facility; data from telemetry.
[d]Accuracy of data is questionable.

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TABLE IV. - PERFORMANCE OF LIFE TEST M THRUSTER

(Item index as in table I)

Item	100% thrust					80% thrust	
	Test 1[a]	Life test, hr					
		10	1000	2000	2740	2750	4500
1	(c)	[c]1.25	[c]1.25	[c]1.25	[c]1.25	[c]1.25	[c]1.25
2	1.86	1.76	1.75	1.76	1.75	1.75	1.75
3	7.0	[c]2.9	[c]3.0	[c]3.1	[c]3.1	[c]2.7	[c]2.7
4	1.5	1.34	1.35	1.35	1.35	1.34	1.35
5	36.9	37.5	37.4	37.4	37.3	37.5	37.5
6	1.85	1.90	2.01	2.03	2.09	1.31	1.32
7	2920	3180	3190	3170	3190	3230	3240
8	0.250	0.255	0.255	0.255	0.255	0.200	0.200
9	1520	1660	1680	1670	1680	1680	1680
10	1.2	1.4	1.2	1.2	1.2	1.6	1.6
11	8.1	[c]3.8	[c]3.5	[c]2.6	[c]2.6	[c]3.0	[c]2.6
12	2.28	1.98	1.92	1.69	1.68	1.76	1.64
13	22.5	22.5	22.3	22.0	22.0	22.1	22.0
14	0.205	0.202	0.203	0.200	0.200	0.204	0.205
15	-14	-15	-16	-18	-20	-18	-18
16	245	255	255	253	254	198	198
17	11.7	11.2	11.3	11.4	11.3	12.2	12.6
18	0.30	(c)	(c)	(c)	(c)	(c)	(c)
19	2×10^{-6}	8×10^{-6}	2×10^{-6}	2×10^{-6}	2×10^{-6}	1×10^{-6}	1×10^{-6}
20	0.310	[b]0.309	[b]0.308	[b]0.308	[b]0.308	[b]0.286	[b]0.286
21	0.023	[b]0.023	[b]0.023	[b]0.024	[b]0.024	[b]0.029	[b]0.029
22	60	66	66	66	66	66	66

[a]Calibration test in 1.5-meter diameter vacuum facility.
[b]Flow rates estimated from previous tests.
[c]Accuracy of data is questionable or data unavailable.

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TABLE V. - PERFORMANCE OF LIFE TEST T THRUSTER

(Item index as in table I)

Item	100% thrust				
	Test 1[a]	Life test, hr			
		10	1000	2000	2886
1	(c)	[c]2.40	[c]2.32	[c]2.12	[c]1.90
2	1.99	1.75	1.72	1.70	1.72
3	[c]4.3	9.0	9.0	9.1	9.0
4	1.49	1.40	1.40	1.40	1.40
5	37.0	36.6	36.4	36.3	36.1
6	1.62	1.67	1.77	1.84	1.95
7	2990	3130	3130	3030	2920
8	0.250	0.252	0.253	0.253	0.252
9	1540	1640	1640	1580	1470
10	1.7	2.6	2.5	2.6	2.3
11	7.8	4.5	4.4	4.6	4.6
12	2.17	1.96	1.98	2.02	2.00
13	22.0	23.0	23.1	23.2	23.0
14	0.200	0.210	0.212	0.207	0.207
15	-17	-20	-22	-20	-21
16	247	(c)	(c)	(c)	(c)
17	12.5	12.5	12.5	(c)	(c)
18	0.304	(c)	(c)	(c)	(c)
19	3×10^{-6}	2×10^{-5}	1×10^{-5}	1×10^{-5}	1×10^{-5}
20	0.315	[b]0.320	[b]0.319	[b]0.319	[b]0.320
21	0.022	[b]0.021	[b]0.021	[b]0.020	[b]0.020
22	60	66	66	64	62

[a]Calibration test in 1.5-meter diameter vacuum facility.
[b]Flow rates estimated from previous tests.
[c]Accuracy of data is questionable or data unavailable.

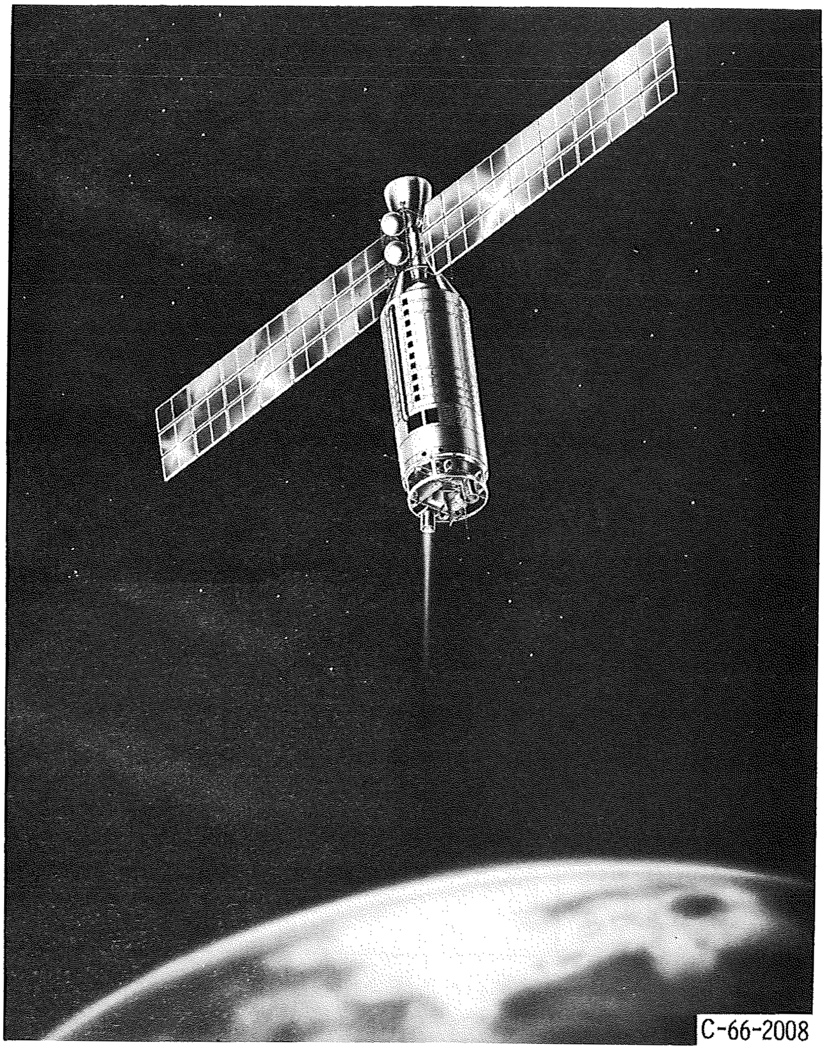

Figure 1. - SERT II spacecraft in orbit (artist's concept).

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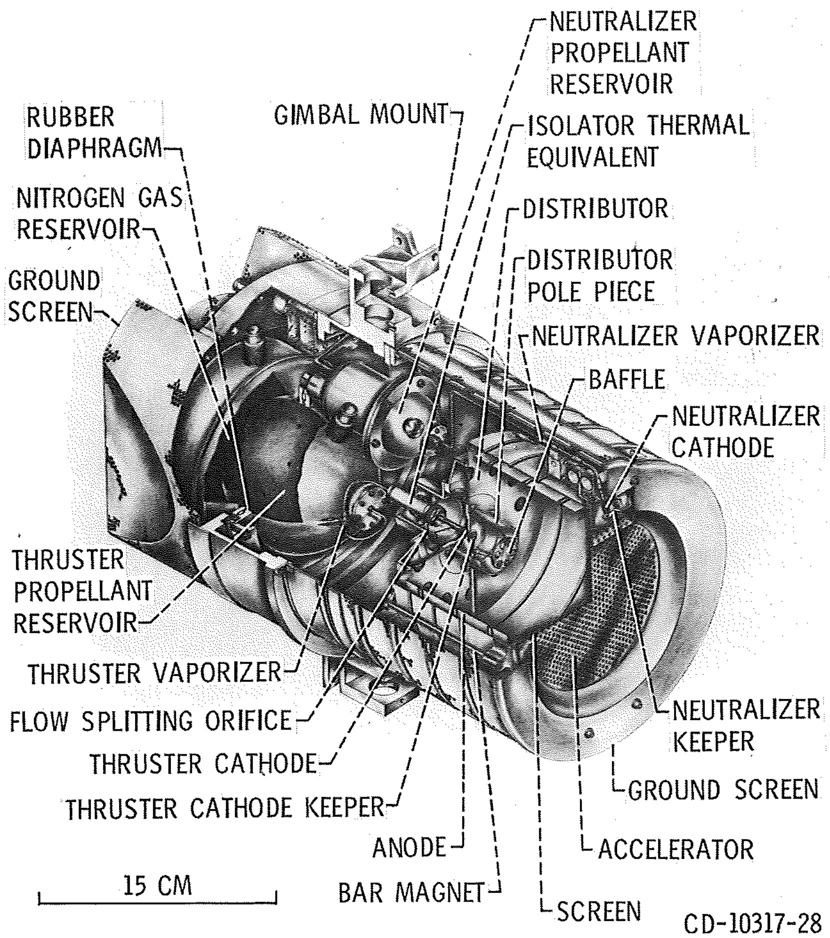

Figure 2. - SERT-II thruster.

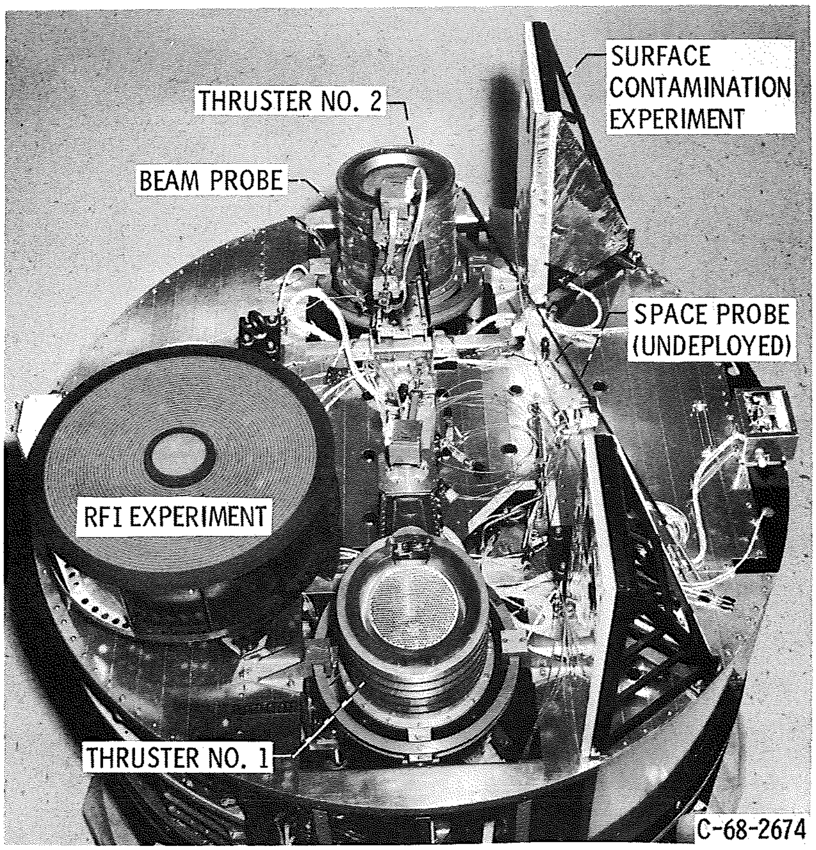

Figure 3. - SERT II flight spacecraft, 1.5 meter diameter.

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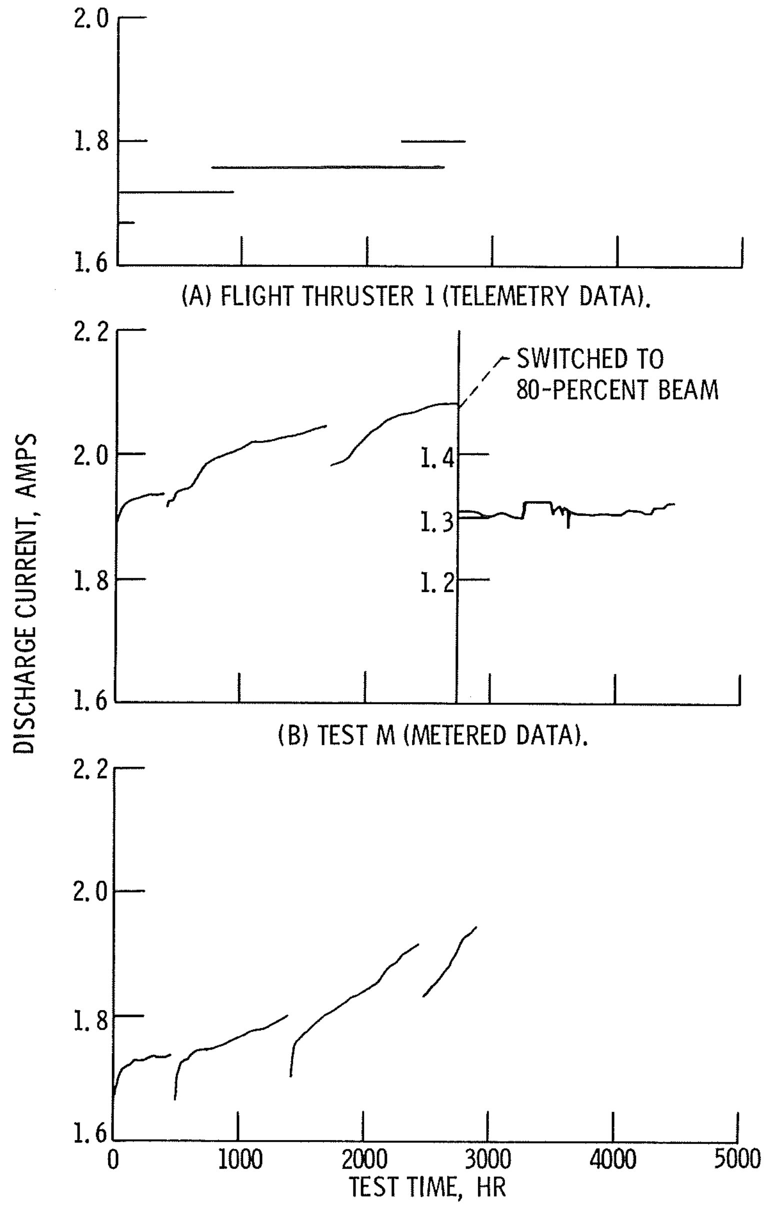

Figure 4. - Trend of discharge current with thruster test time at full beam. See tables I, IV, or V for complete test conditions.

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